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Final Report

SOME NUMERICAL SOLUTIONS OF INVISCID, UNSTEADY, TRANSONIC FLOWS OVER THE NLR 7301 AIRFOIL

R. J. Magnus

Convair Division of General Dynamics
San Diego, California

January 1978

Sponsored by the
OFFICE OF NAVAL RESEARCH (Code 438)
Department of the Navy

Contract No. N00014-77-C-0051
ONR Contract Authority No. NR 061-214

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FOREWORD

This research was undertaken by the Convair Division of General Dynamics, P. O. Box 80847, San Diego, California 92138. The work was done under the Office of Naval Research, Contract N00014-77-C-0051, ONR Contract Authority NR 061-214/10-21-76 (438). The ONR scientific officer was Mr. Ralph D. Cooper, Director, Fluid Dynamics Programs and the monitor of the technical effort was Mr. Morton Cooper. Dr. R. J. Magnus carried out the computer programming and the calculations. The assistance and cooperation of Dr. H. Lomax, Dr. W. Ballhaus, and Dr. Sanford Davis at NASA Ames Research Center are acknowledged. The cooperation of Dr. H. Tijdeman and others at the NLR in suggesting this project, in furnishing the airfoil description, and in furnishing reports on their experimental work for guidance in the conduct of this project are greatly appreciated.

ABSTRACT

Inviscid transonic flows over the NLR 7301 airfoil were calculated with a program based on the unsteady Euler equations. The blunt-nosed, 16.5 percent thick, aft-cambered section is of the type designed for shock-free flow under prescribed conditions. Steady flows were calculated for four Mach number-incidence combinations (0.500 @ 0.85°, 0.700 @ 3.00°, 0.721 @ 0.00°, 0.744 @ 0.85°) for the airfoil in an unrestricted stream; also at Mach 0.744 @ 0.85° incidence for the airfoil in a slotted-wall tunnel and in a free-jet. Quasi-steady behavior was checked by calculating steady flows at incidences ±0.50° from the basic incidence mentioned. Unsteady flows were calculated with the airfoil pitching ±0.50° about an axis at 0.40 chord at reduced frequencies ($k \equiv \omega C/2U_{\infty}$) on the order of 0.2; the actual frequency for each case was chosen to duplicate the 80 hertz maximum oscillation rate achieved in tests of this airfoil by Tijdeman.

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SECTION I

INTRODUCTION

The work of Tijdeman, References 1 - 7, on unsteady flow has, obviously, been the inspiration for a revival of interest in transonic unsteady flow over airfoils. There have been many methods proposed for calculating such flows and, by and large, they have been exercised on the problem of flow over the 64A006 with oscillating flap, a configuration tested by Tijdeman. This configuration appears ideal for testing the usefulness of some of the simpler analytic and numerical methods -- thin airfoil, small inclinations and low lift. Analogous calculations of the unsteady flow over the NLR 7301 supercritical airfoil, also tested by Tijdeman, have not appeared in the literature. The present work is an application of an inviscid numerical procedure to the analysis of unsteady flow over this 16.5 percent thick, blunt-nosed, highly-cambered section.

Because the quasi-steady lift curve slope, $d\,C_L/d\alpha$, for a supercritical section becomes large for conditions which result in shockless supercritical flow on the upper surface, the unsteady characteristics in pitching are of particular interest. Apprehensions that overly severe unsteady conditions might develop in oscillating an airfoil in pitch about a "shockless" state largely have been dispelled by Tijdeman's experiments and the calculations of Isogai, who studied a Bauer-Garabedian-Korn-Jameson section, Reference 8.

The difficulties in calculating transonic flows over modern supercritical sections and in comparing the results with wind-tunnel data were discussed by Kacprzynski, Reference 9. Some progress on handling the important viscous effects by boundary layer techniques has been reported by Bauer, et al, Reference 10, and by Melnik and Mead, Reference 11. Also, viscous effects being treated by use of the "Navier-Stokes" approach are reported, for example,

by Walitt, King and Liu, Reference 12 and by Rose and Seginer, Reference 13. Thus, the present inviscid calculations should be regarded as a (possibly) severely approximate means of analysis of unsteady flows over real supercritical sections.

A number of features or characteristics of the present method might be regarded as favorable in analyzing the flow over a thick airfoil. The method uses an explicit Lax-Wendroff scheme to obtain numerical solutions to the unsteady Euler equations in conservative form. Discontinuities in solutions to these equations have the jump properties of ordinary gas dynamic shocks and the scheme captures these discontinuities. Thus, total head loss through stronger shocks is properly accounted for. The calculation method uses several mesh systems including fixed outer systems and local systems attached to the moving airfoil surface. Terms are added to the equations used with the local accelerating coordinates to maintain conservative form and the tangency boundary conditions are satisfied at nodes on the moving airfoil surface. If desired, tunnel wall boundary conditions can be satisfied at nodes in the fixed, outer coordinate systems.

Other features may be regarded unfavorably. There are no viscous terms in the equations, so decambering because of boundary layers near the trailing edge is not simulated and lift coefficients may be too large. Also, because interaction of shocks with boundary layers on the airfoil surfaces are not accounted for, the calculated shocks may be too strong, too far aft, and not move properly when incidence is varied. High resolution and low computing cost are antithetical; there is evidence that not enough mesh was used in the region of the airfoil nose in the present work. To maintain stability in the numerical process, numerical diffusion has been added which degrades the sharpness of flow gradients in the solutions. Usage of the four coupled equations throughout the flowfield is unnecessary; the flow which lies ahead of shocks or on lines which do not penetrate shocks ought to be describable by simpler equations. Solving the system with an explicit scheme causes most of

the computer usage to be expended in detailing the finer mesh parts of the flow (around the airfoil nose for example) and makes the program costly to run. This expense makes it unattractive to calculate low reduced frequency cases or to run large numbers of examples to establish trends of airfoil properties as Mach number, incidence, or motion frequency and amplitude parameters are varied.

SECTION II

COMPUTATIONAL NOTES

The computer program used in the present work is a modification of the program used in previous calculations of unsteady flows over the NACA 64A006 airfoil, Reference 14, and is closely related to the program used in work on the NACA 64A410, Reference 15.

The coupled system of four unsteady Euler equations in conservation form is solved numerically using a two-step, Lax-Wendroff, explicit, finite-difference scheme. Diffusion was added to suppress ragged overshoot in the calculated output near shocks and was also found to be needed to control short-wavelength oscillations in parts of the flow which were near-sonic.

On the order of 5000 mesh nodes arranged in several distinct grid systems were used to cover the field around the airfoil. Fine mesh was used around the airfoil nose; the basic mesh around the airfoil was 0.04 chord squares, and stretched and coarser meshes were used to extend the coverage to outer boundaries several chords from the airfoil.

Local, airfoil-oriented, moving coordinate systems two or three cells deep were used to provide mesh nodes along the moving airfoil surface. The conservative form for the flow equations in these accelerating coordinates was provided by the method outlined by Viviand, Reference 16. Fixed underlying mesh was used to facilitate satisfying boundary conditions along lines representing wind-tunnel walls if desired. Exchanges of information between the developing solutions in the fixed and moving systems were made by interpolations. That the airfoil would make only small excursions (± 0.5 degree pitch) with respect to the fixed background was utilized to simplify the interpolation logic.

More fine mesh was needed to detail the flow around the blunt nose of the NLR 7301 than was used in calculating the relatively sharp-nosed NACA 64A006. Using fine mesh is costly when an explicit scheme is used because the allowable time step (limited by computational instabilities) is dependent on the mesh size. To prevent the program from becoming too costly to run, fine mesh was used more sparingly over some of the aft parts of the airfoil than had been used for calculating the 64A006. Inspection of the results of calculating flows over the NLR 7301 indicates that the mesh around the nose may have been too coarse to yield a good approximation to the "shockless" flow state.

To satisfy tangency boundary conditions along the airfoil surface, the flow at nodes on the moving surface is calculated by the method cataloged as "Euler Predictor, Simple Wave Corrector" in the survey by Abbett, Reference 17. By a similar process the upper and lower pressures and flow directions are matched along a line extending aft about 0.2 chord from the airfoil trailing edge. Further aft the wake discontinuity is allowed to become indistinct by numerical diffusion.

If the flow over the airfoil in an unrestricted stream was to be calculated the flow was held invariant at the field perimeter. On the examples calculated here, the upstream and downstream field boundaries were 7.0 chords distant from the airfoil midchord and the lateral boundaries were placed at 10.4 chords by use of stretched mesh. A flow pattern due to a doublet and a vortex (strength commensurate with airfoil mean lift) plus free stream was maintained on the perimeter.

If free-jet conditions were to be simulated, uniform free stream pressure was maintained along horizontal lines ±1.53 chords from the airfoil and along the downstream boundary. At the upstream boundary the flow was specified to be uniform free stream.

For the cases simulating flow over the airfoil in a slotted wall tunnel, an empirically determined boundary condition on perturbation velocities

$$u \pm 0.73v \pm 0.17v_x = 0$$

was specified on horizontal lines ±1.53 chords from the airfoil. The basis for this empirical relation is discussed in Reference 14.

The computer programs utilized are written in FORTRAN extended language and the calculations were run on a CDC 7600 computer. A relatively stable solution to a steady flow problem would be obtained in about 2400 passes through the field requiring 580 seconds of computation. Stationary solutions to the unsteady problems typically would be obtained after following the flow for about 3.5 cycles; this would require 2200 seconds of computing. Pressure fields at (typically) 48 steps in an oscillation cycle were recorded for further study.

SECTION III

CALCULATED EXAMPLES

1. Airfoil Shape

The NLR 7301 airfoil being studied is a 16.5 percent thick supercritical section designed for shockless operation. Experimentally, shockless flow occurs at a lift coefficient of about 0.46 at Mach 0.75, Reference 7. The general appearance of the airfoil and the coordinates, as listed in Reference 7, are shown in Figure 1.

2. Basic Steady Flows

Using the explicit finite difference program the steady flow over the NLR 7301 airfoil was calculated for six basic conditions, the integrated results are listed in Table 1.

At Mach 0.500 and 0.85 degrees angle of attack, Figure 2, the flow is subsonic over the entire surface. At Mach 0.700 and 3.00 degrees angle-of-attack, Figure 3, the flow is supersonic over a considerable part of the upper surface and the supersonic region terminates at a strong shock near 0.63 chord. The Mach number ahead of the shock is about 1.48, considerably larger than what could be tolerated in a real flow without separating the boundary layer. At Mach 0.721 and zero angle of attack, Figure 4, the upper surface flow is supercritical but near a shockless state. The three basic cases mentioned above were all calculated assuming the airfoil to be immersed in an unrestricted stream.

At Mach 0.744 and 0.85 degrees angle-of-attack the steady flow was calculated assuming the airfoil to be immersed in an unrestricted stream, in a slotted wall tunnel, and in a free jet. The tunnel height and free-jet

height were both assumed to be 3.06 chords. In an unrestricted stream, Figure 5, there is a strong shock ($M_1 \doteq 1.5$) near 0.71 chord on the upper surface. In the slotted wall tunnel and free jet environments, Figures 6 and 7, there is a strong compression or a supersonic-to-supersonic shock between 0.15 and 0.20 chord and a relatively weak supersonic-to-subsonic compression near 0.7 chord on the upper surface.

The upper surface flow patterns seen in these cases are fairly characteristic of the variety of patterns to be expected in inviscid flow over this airfoil. All steady flow cases having unrestricted stream outer boundary conditions described above were also calculated using a Bauer-Korn program, Reference 10, based upon numerical solutions of the potential flow equation; the non-conservative shock capturing scheme was utilized. Basic cases as calculated using the Bauer-Korn program are also shown in Figures 2 through 5. At Mach 0.500 the two methods of solution agree fairly well. However, a detailed comparison shows that the expansion of the flow along the upperside of the nose occurs too slowly in the solutions generated by the present program. This tendency is exaggerated in the solutions with higher lift and Mach number; see Figures 3 through 5.

Insufficient expansion of the flow along the upper part of the nose was noted early in the project and diagnosed as being due to use of too coarse a mesh in the nose region. The program was altered to provide more mesh but the tendency toward insufficient expansion was not completely overcome. Diagnostic checks were made whereby the surface pressures generated by the Bauer-Korn program were assigned as boundary conditions at points 2 to 9% chords aft along the nose but these modifications failed to change pressures calculated by the explicit-finite difference program for points further aft along the upper surface. It is surmised from this result that expansion waves emanating from the upper nose region probably are being attenuated by too coarse a mesh in some region outboard of the

surface. The distortions to upper surface pressure distributions occasioned by this shortcoming of the present program are most apparent for the case at Mach 0.721 for which the flow is near shockless, see Figure 4. It should be assumed here that the Bauer-Korn program provides a superior solution.

The Mach numbers selected for the calculations presented in Figures 2, 3, and 5 are the same as those in tests run by Tijdeman, Reference 7, and the incidences selected for the calculations match the geometric incidences set in Tijdeman's experiments. Because the calculations in Figures 2, 3 and 5 do not account for viscous effects or wind tunnel wall influence, they indicate much more lift than the correspondent basic experimental steady flews published by Tijdeman, Reference 7.

The inclusion of a homogeneous slotted wall boundary condition in the calculations at Mach 0.744, Figure 6, drops the lift to about half of the value calculated for the airfoil at the same geometric incidence in an unrestricted stream, Figure 5. The lift and the pressure distribution shown in Figure 6 indicate an effective incidence too low for a good match with Tijdeman's results, Reference 7. Of course, it should not be forgotten that the slotted wall boundary condition included in the calculation is speculative and not based upon measurements from Tijdeman's experiments.

3. Quasi-Steady Perturbations

The lifts and moments resulting from steady flow calculations at incidences of 0.5 degree above and below the basic incidences are also listed in Table 1. The results from comparable calculations using the Bauer-Korn program (with non-conservative shock capturing) are listed in Table 2.

At Mach 0.500 and Mach 0.721, for which the shocks are either weak or absent, the lifts calculated by the two programs agree relatively well as to their changes in a one degree incidence range about the basic incidence. At Mach 0.744, however, the lift change for 1.0 degree incidence change is only about 0.88 of the lift change calculated by the Bauer-Korn program. At Mach 0.700 and 3.0 degrees incidence the present program shows a lift change due to 1.0 degree incidence change which is only about 0.75 of the change predicted by the Bauer-Korn program.

Certainly the two programs treat shocks differently; however, other mechanisms might account for the discrepancies between the calculated $\Delta C_L/\Delta\alpha$ values. The "tightness" of the numerical procedures by which the Kutta conditions are enforced in the two programs might differ; however, that the results from the programs agree quite well for cases with weak shocks makes this suggestion unlikely. For the problems at Mach 0.700 and 0.744 the Mach number just ahead of the shock is of the order of 1.5; the program based upon the Euler equations, therefore, calculates that the fluid washing the upper side of the trailing edge has suffered (roughly) seven percent loss of total head. The Bauer-Korn program, being based on the potential equation, would not account for altered properties for this stream. Possibly, because of the lowered total head of the upper surface stream the present program would allow more (concave upward) curvature on the dividing streamline than does the Bauer-Korn program.

Chordwise distributions of the normalized quasi-steady pressure loading are shown in Figures 8 through 13. Here ΔC_p is arbitrarily defined as:

$$\Delta C_p = 57.3\,(C_{p+} - C_{p-})$$

where C_{p+} is the pressure coefficient at the nominal incidence plus 0.5 degree and C_{p-} is the pressure coefficient at nominal incidence minus 0.5 degrees.

For those cases having the airfoil in an unrestricted stream, Figures 8 through 11, quasi-steady pressure changes as calculated by the Bauer-Korn program are also presented. Differences between the results produced by the two methods may be noted. On the upper surface near the nose, the superior detailing of the flow expansion by the Bauer-Korn program is evident.

When a strong shock is present, Figures 9 and 11, it is evident that the differences in lift slope calculated by the two methods (noted earlier) are not caused solely by different pressures on the upper surface aft of the shock; the loadings on the lower surface and upper surface forward also are not in agreement.

In shape, these distributions bear some resemblance to the experimental results reported by Tijdeman, References 6 and 7. Since none of the basic, steady-flow, pressure distributions matches any of Tijdeman's, the changes due to ±0.5 degree incidence do not match either. The spikes in loading due to shock movement are more intense for the inviscid calculations than for the experiments because the computations do not account for shock weakening by interaction with the boundary layer.

4. Unsteady Perturbations

Unsteady oscillatory flows for the six basic cases were calculated and the results are presented in Table 3 and Figures 14 through 19. The reduced frequencies chosen are those which would have prevailed in Tijdeman's experiments, References 6 and 7, if 80 hertz frequency were selected.

For these cases the airfoil was assumed to oscillate in pitch about an axis 0.40 chord aft of the airfoil nose and 0.017 chord below the airfoil reference chordline. Sinusoidal incidence variations of 0.5 degree around the basic incidences were assumed:

$$\alpha(t) = \alpha_0 + 0.5 \sin \omega t.$$

The reduced frequency and circular frequency are related as follows:

$$k = \omega C/2U_{\infty}.$$

The oscillatory lifts, pitching moments, and pressures at selected locations along the airfoil surfaces were fitted with three harmonic representations.

$$F(t) = \bar{F} + \sum_{n=1}^{3} (R_n \sin n\omega t + I_n \cos n\omega t).$$

Mean values and the real and imaginary parts of the first harmonics of the lifts and pitching moments are listed in Table 3. For the cases calculated, the magnitudes of the second harmonics of the lift were all less than 3 percent of the magnitudes of the corresponding first harmonics. The second harmonics of the pitching moments ranged between 4 and 16 percent of the magnitudes of the fundamentals; 4 percent for the subsonic example at Mach 0.50 and 16 percent for the case at Mach 0.744 in a free jet.

The real and imaginary parts of the normalized first harmonics of the surface pressure excursions for the various cases calculated are shown in Figures 14 through 19. The responses presented are the magnitudes of the excursions of pressure coefficient per radian of pitch oscillation amplitude. Of course, the height and broadness of each pressure spike caused by shock motion only makes sense if it is recalled that the actual amplitudes of the pitch oscillations for the calculations was 0.50 degrees.

In general, the second harmonic of oscillatory pressure would be less than 10 percent of the magnitude of the corresponding fundamental. Exceptions to this rule occurred near or aft of a strong shock and on the forward part of the upper surface of the airfoil for the "shockless" case at Mach 0.721; the quasi-steady pressure changes for positive and negative incidence changes also would be unequal (non-linear) for the conditions mentioned.

SECTION IV
DISCUSSION

The present calculations of unsteady transonic flows over the NLR 7301 supercritical section have been motivated by the existence of Tijdeman's experimental work on this airfoil. The schedule of inviscid calculations also is patterned on Tijdeman's work:

a. Select and calculate a number of basic cases which demonstrate a variety of upper surface flow patterns.

b. Calculate steady flows at incidences ±0.5 degrees on either side of the basic incidence to assess quasi-steady behavior.

c. Calculate the unsteady flow for the airfoil oscillating sinusoidally in pitch ±0.5 degree about an axis at 0.40 chord

Because the calculations do not include viscous effects and because wind-tunnel wall effects have been included for only one Mach number and incidence in a speculative manner, comparisons with Tijdeman's experimental data have not been emphasized. Note that the calculated flow patterns (as a collection of cases) have many of the features so aptly described by Tijdeman, Reference 7.

Compared to the experimental results, the calculated results may show some differences which can be expected on logical grounds:

a. Larger pressure loading spikes on parts of the surface traversed by shocks because shock weakening by interaction with the boundary layer has not been included --

b. Altered basic shock placement, shock movement with incidence changes, and altered pressure distributions on the airfoil surfaces aft of the shocks, also because the calculated shocks are too strong --

c. Pressures on the aft lower surface of the airfoil more positive because the thickening of the boundary layer in the strong adverse pressure gradient ahead of the lower surface concavity has not been accounted for.

Assessing wind tunnel wall effects on the unsteady pressures, per se, was not accomplished. The changes in basic steady flow patterns due to changing from an unrestricted stream to a slotted-wall or a free-jet outer boundary were so drastic (C_L change from 0.81 to 0.41 or 0.33 respectively) that seeking any subtle differences in unsteady pressures on the airfoil due to different "returns" from the outer boundary would be useless.

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Table 1. NLR 7301 Airfoil, Steady Forces and Moments, 0.5 Degree Perturbations About Nominal Angle-Of-Attack Inviscid (Euler) Program

Coefficients:

$C_L =$ Airfoil Lift/qC

C_m = Airfoil nose-up moment about quarter chord/qC^2

C = chord

q = Free stream dynamic pressure

$\Delta\alpha$ = 0.5 degree

Outer Boundary Condition	Mach	α_o	Lift Coefficients $\alpha_o-\Delta\alpha$	Lift Coefficients α_o	Lift Coefficients $\alpha_o+\Delta\alpha$	Pitching Moment Coefficients $\alpha_o-\Delta\alpha$	Pitching Moment Coefficients α_o	Pitching Moment Coefficients $\alpha_o+\Delta\alpha$
Unrestricted Stream	0.500	0.85	.4393	.5152	.5881	-.1009	-.1021	-.1028
Unrestricted Stream	0.700	3.00	1.0828	1.1601	1.2354	-.1438	-.1528	-.1627
Unrestricted Stream	0.721	0.00	.4335	.5499	.6891	-.1246	-.1230	-.1257
Unrestricted Stream	0.744	0.85	.6982	.8057	.9040	-.1529	-.1670	-.1822
Homogeneous Slotted Walls @ y = ±1.53C	0.744	0.85	.3493	.4112	.4764	-.1268	-.1277	-.1284
Free Jet Surfaces @ y = ±1.53C	0.744	0.85	.2825	.3251	.3761	-.1132	-.1099	-.1077

Table 2. NLR 7301 Airfoil, Steady Forces and Moments, 0.5 Degree Perturbations About Nominal Angle-Of-Attack Bauer-Korn (Potential Flow) Program

Coefficients:

$C_L =$ Airfoil Lift/qC

$C_m =$ Airfoil nose-up moment about quarter chord/qC^2

C = chord

q = Free stream dynamic pressure

$\Delta\alpha = 0.5$ degree

			Lift Coefficients			Pitching Moment Coefficients		
Outer Boundary Condition	Mach	α_0	$\alpha_0-\Delta\alpha$	α_0	$\alpha_0+\Delta\alpha$	$\alpha_0-\Delta\alpha$	α_0	$\alpha_0+\Delta\alpha$
Unrestricted Stream	0.500	0.85	.4317	.5068	.5821	-.0963	-.0958	-.0951
Unrestricted Stream	0.700	3.00	1.0936	1.2002	1.2966	-.1467	-.1625	-.1787
Unrestricted Stream	0.721	0.00	.4465	.5665	.6928	-.1267	-.1276	-.1309
Unrestricted Stream	0.744	0.85	.6939	.8091	.9288	-.1570	-.1718	-.1913

Table 3. NLR 7301 Airfoil, Unsteady Forces and Moments In 0.5 Degree Oscillations About 0.4 Chord Axis Inviscid (Euler) Program

Angle of Attack:

$\alpha(t) = \alpha_o + 0.5° \sin(\omega t)$

ω = oscillation rate, radian/time unit

Reduced Frequency:

$k = \frac{\omega C}{2U_\infty}$

C = chord

U_∞ = Free stream velocity

Coefficients:

C_L = Airfoil Lift/qC

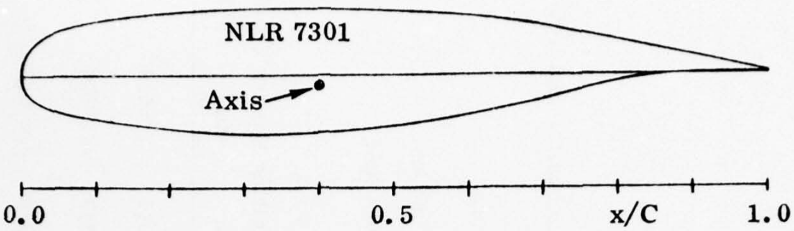
C_m = Airfoil nose-up moment about quarter chord/qC^2

q = Free stream dynamic pressure

Typical First Harmonic of Response Function:

$C_L(t) = \bar{C}_L + Re_L \sin(\omega t) + Im_L \cos(\omega t)$

				Lift Coefficients			Pitching Moment Coefficients		
Outer Boundary Condition	Mach	k	α_o	$\bar{C}_L$	Re_L	Im_L	$\bar{C}m$	Re_m	Im_m
Unrestricted Stream	0.500	0.263	0.85	0.5141	.04691	-.00151	-.1017	-.00023	-.00379
Unrestricted Stream	0.700	0.192	3.00	1.1568	.04728	-.01427	-.1519	-.00504	.00123
Unrestricted Stream	0.721	0.189	0.00	0.5536	.04691	-.02211	-.1232	-.00266	-.00508
Unrestricted Stream	0.744	0.181	0.85	0.8050	.04450	-.01734	-.1671	-.00322	.00220
Homogeneous Slotted Walls @ y = ±1.53C	0.744	0.181	0.85	0.4148	.05642	-.00822	-.1282	-.00019	-.00489
Free Jet Surfaces @ y = ±1.53C	0.744	0.181	0.85	0.3296	.05021	.00423	-.1106	.00236	-.00600



Coordinates of NLR 7301*

Upper Surface x/C	Upper Surface z/C	Lower Surface x/C	Lower Surface z/C
0.0000	-.0004		
.0033	.0196	.0018	-.0134
.0124	.0369	.0079	-.0247
.0207	.0454	.0180	-.0340
.0299	.0511	.0370	-.0437
.0397	.0554	.0650	-.0525
.0499	.0590	.1000	-.0598
.0600	.0618	.1300	-.0643
.0748	.0651	.1649	-.0685
.0998	.0697	.2000	-.0718
.1300	.0741	.2499	-.0750
.1649	.0781	.2998	-.0767
.1995	.0813	.3499	-.0770
.2498	.0847	.3999	-.0760
.2998	.0869	.4497	-.0733
.3497	.0881	.4998	-.0684
.3993	.0883	.5496	-.0613
.4492	.0876	.5996	-.0526
.4996	.0860	.6393	-.0447
.5493	.0832	.6791	-.0361
.5993	.0792	.7193	-.0273
.6493	.0736	.7597	-.0185
.6993	.0661	.7994	-.0104
.7494	.0573	.8377	-.0039
.7982	.0475	.8785	.0013
.8385	.0388	.9188	.0043
.8786	.0297	.9487	.0047
.9184	.0207	.9781	.0037
.9479	.0140		
.9784	.0074		
1.0000	.0030		

*From Reference 7

Figure 1. Shape of the NLR 7301 Airfoil

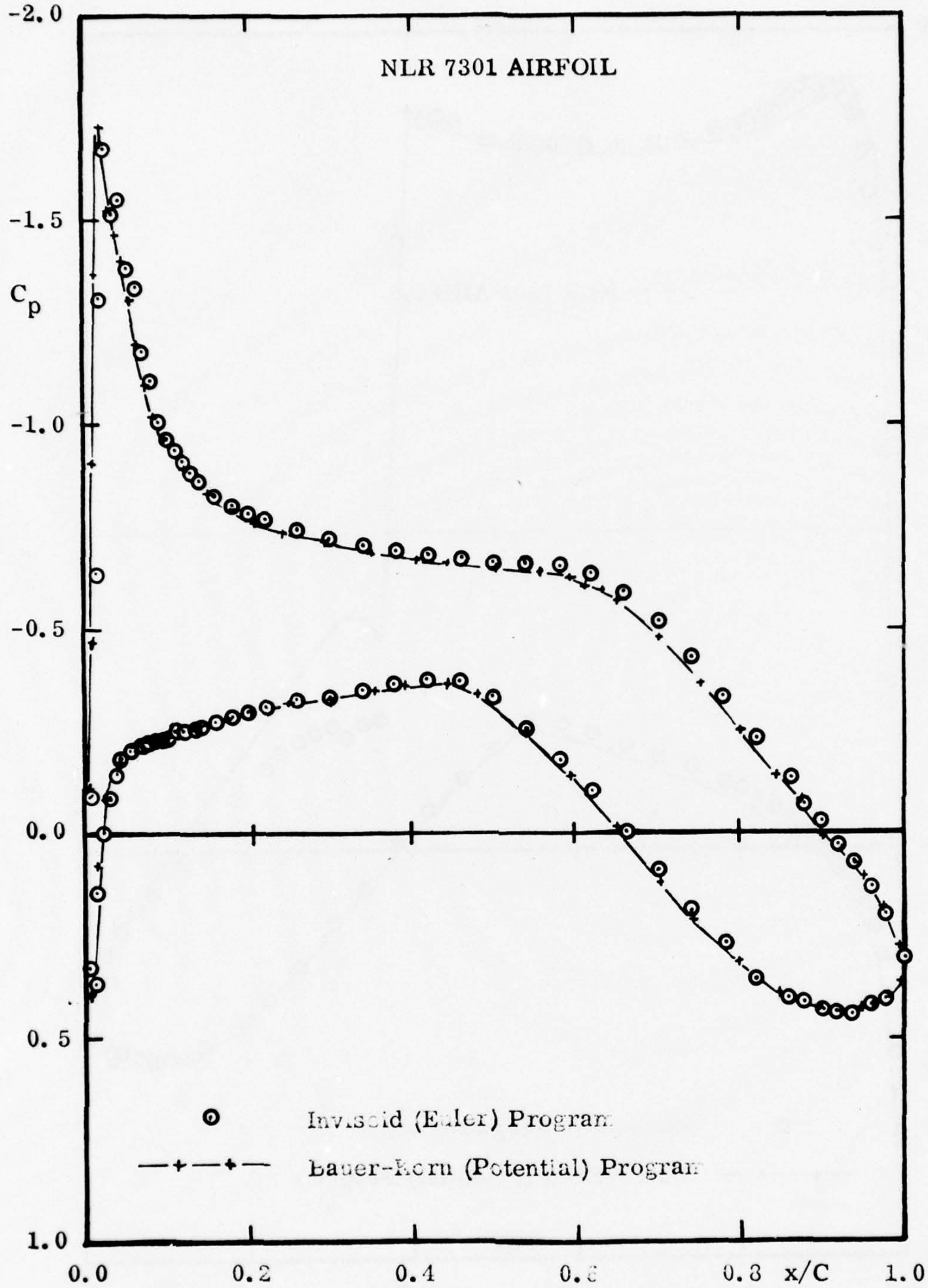


Figure 2. Calculated Pressure Distribution, Mach 0.500, $\alpha = 0.85°$, Unrestricted Stream

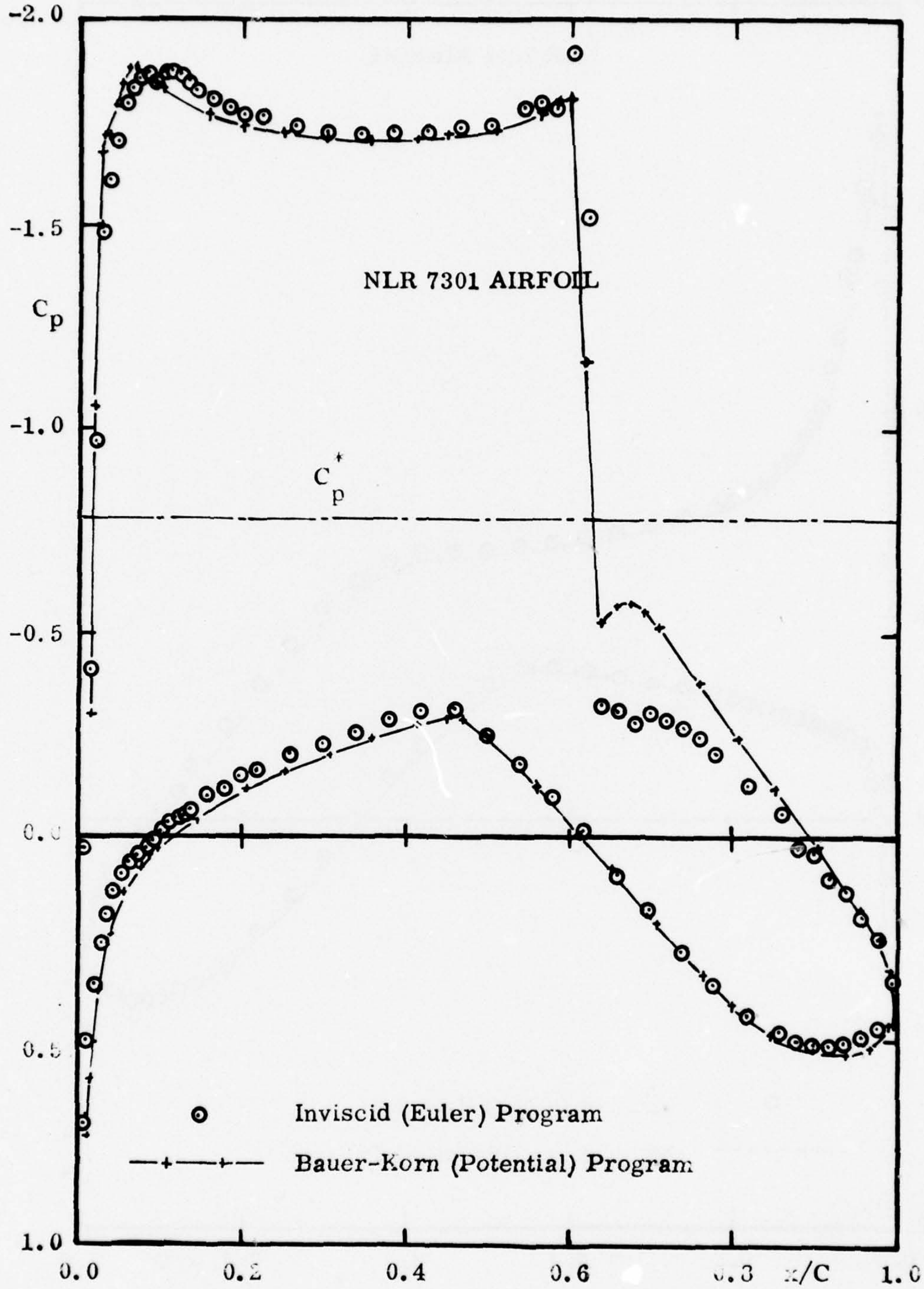


Figure 3. Calculated Pressure Distribution, Mach 0.700, $\alpha = 3.00°$, Unrestricted Stream

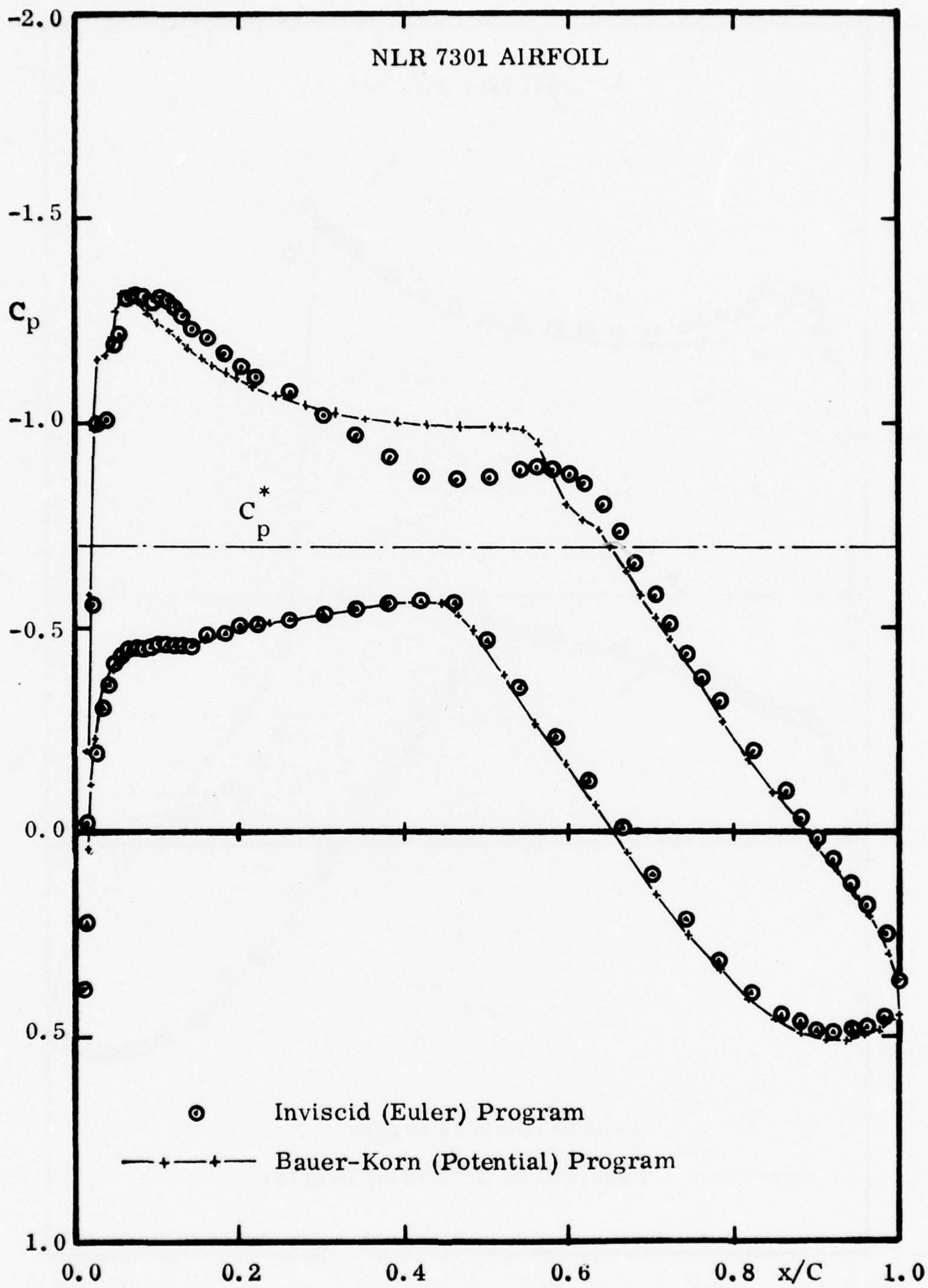


Figure 4. Calculated Pressure Distribution, Mach 0.721, α = 0.00°, Unrestricted Stream

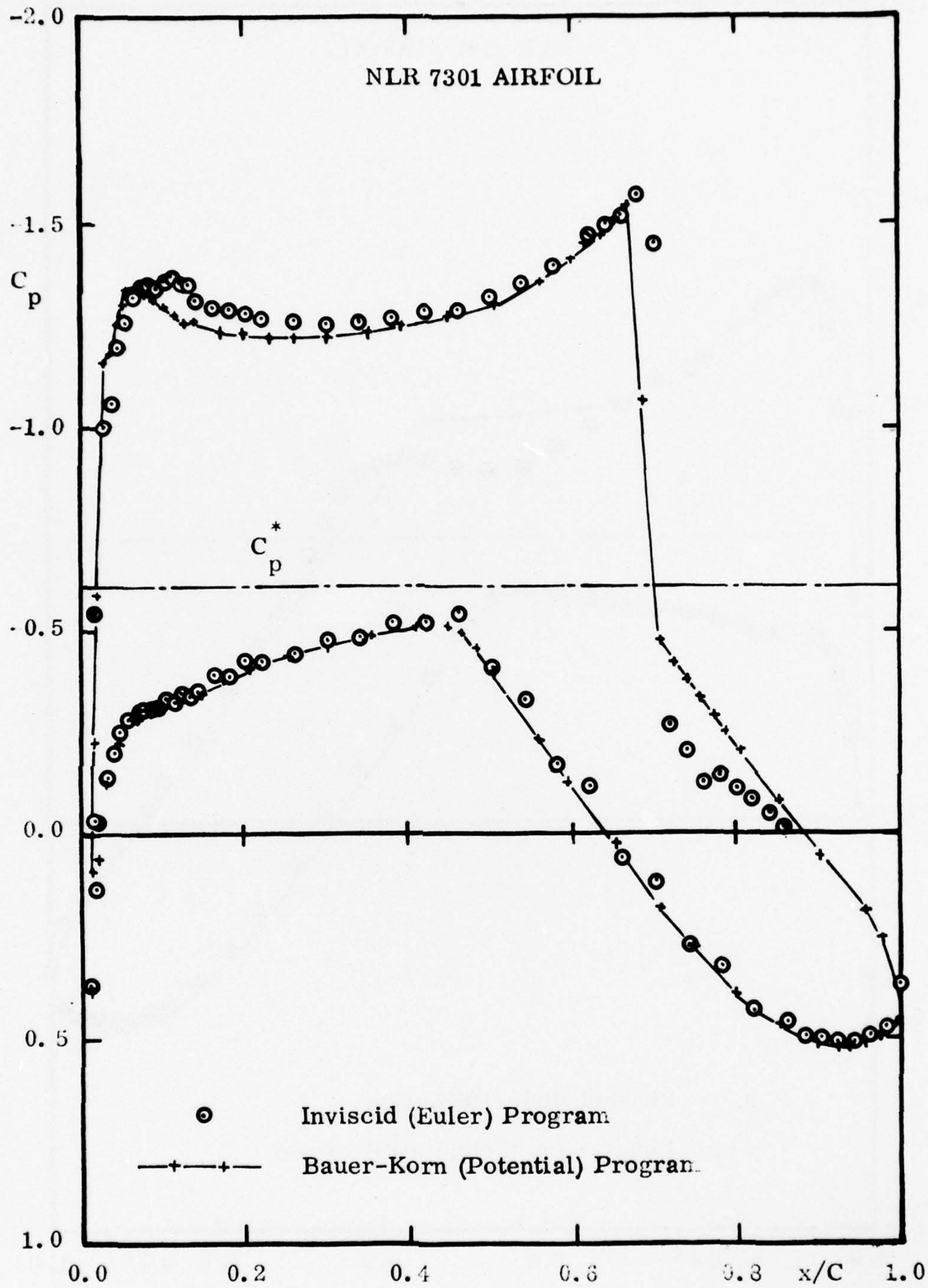


Figure 5. Calculated Pressure Distribution, Mach 0.744, $\alpha = 0.85°$, Unrestricted Stream

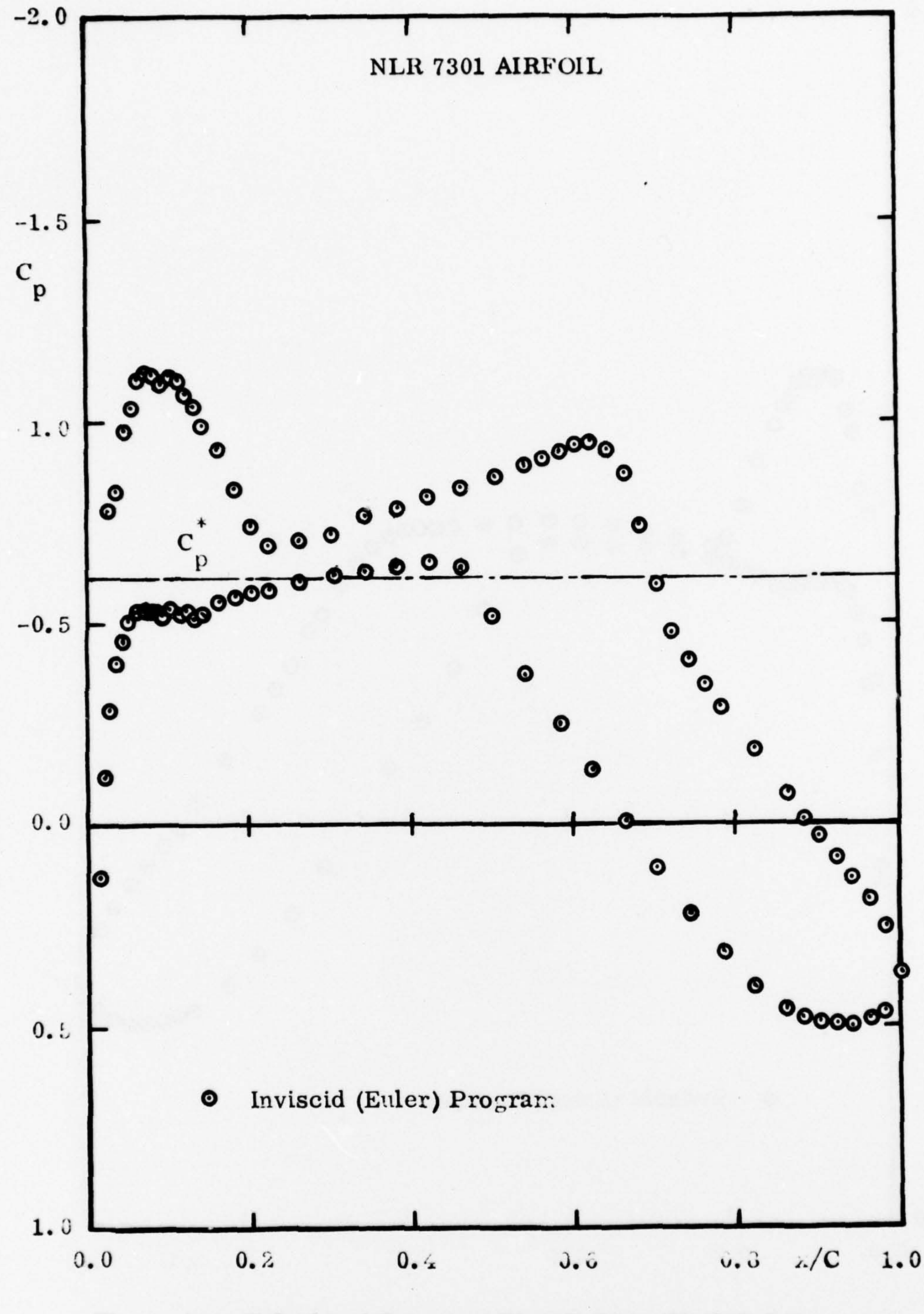


Figure 6. Calculated Pressure Distribution, Mach 0.744, $\alpha = 0.85°$, Slotted Tunnel Walls at $y = \pm 1.53$ Chords

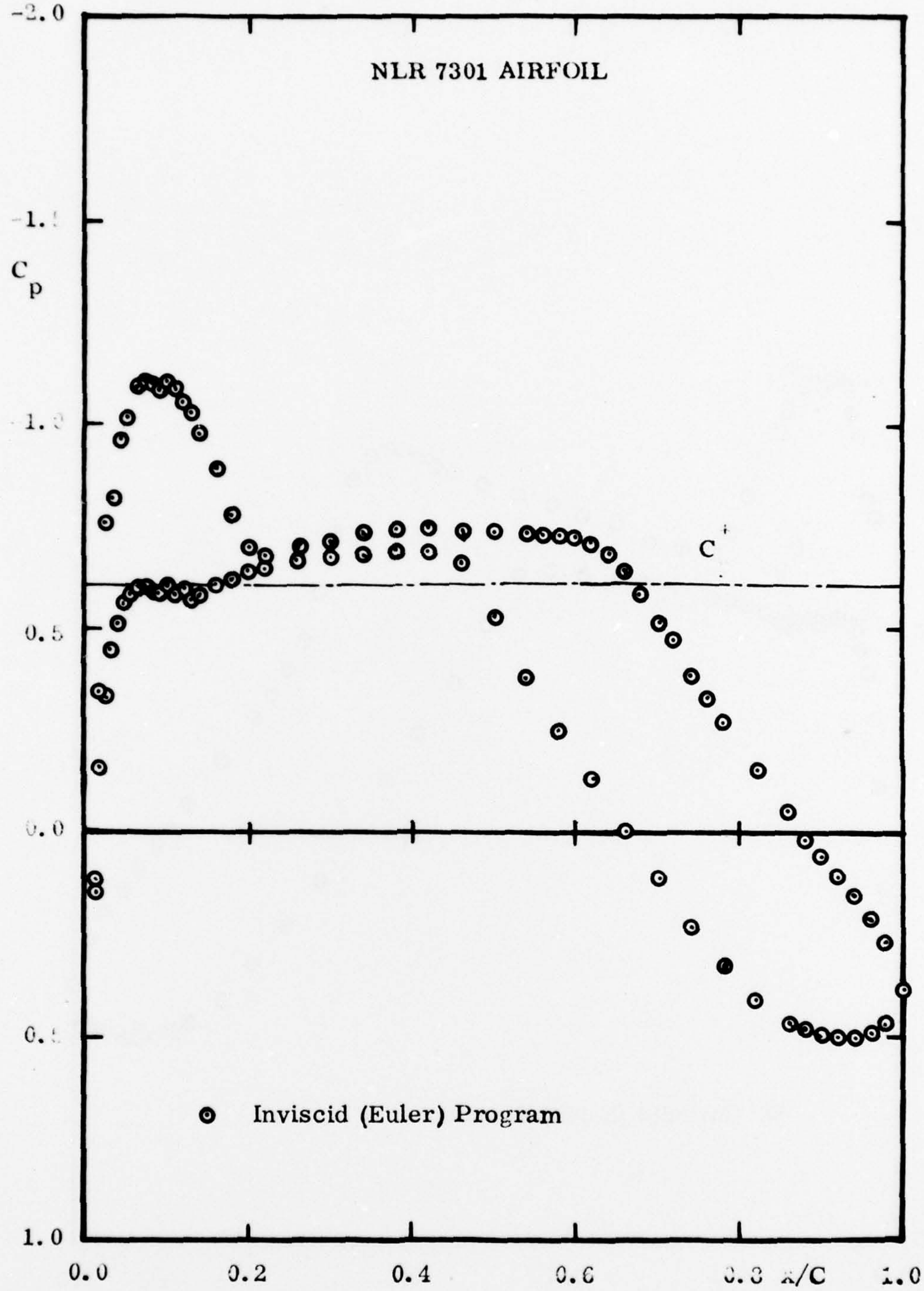


Figure 7. Calculated Pressure Distribution, Mach 0.744, $\alpha = 0.85°$, Free Jet Surfaces at y = ±1.53 Chords

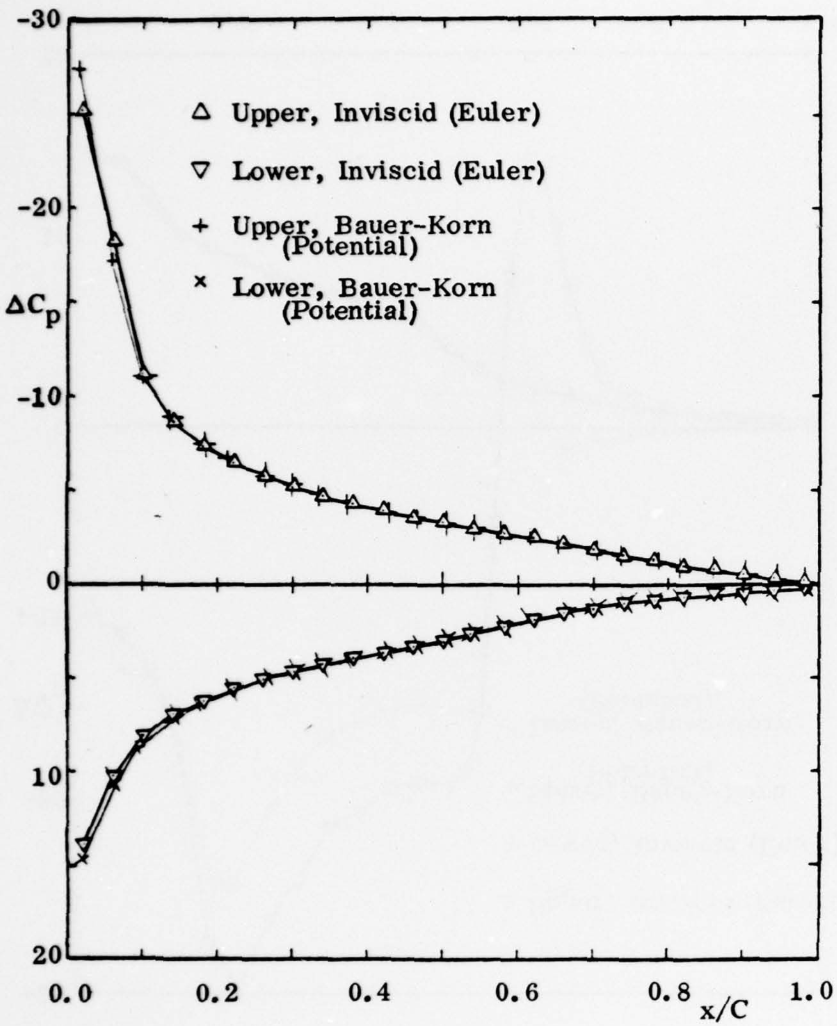


Figure 8. Pressure Excursions in Quasi-Steady Flow, Mach 0.500, $\alpha = 0.85°$, Unrestricted Stream

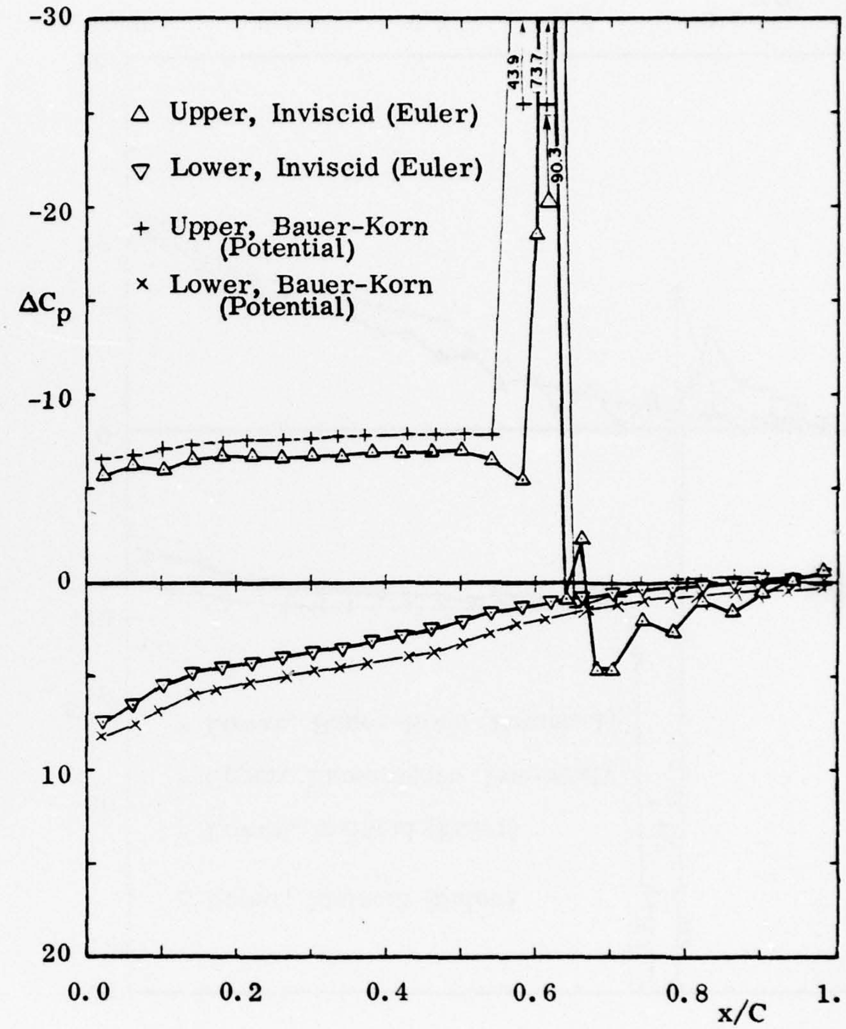


Figure 9. Pressure Excursions in Quasi-Steady Flow, Mach 0.700, $\alpha = 3.00°$, Unrestricted Stream

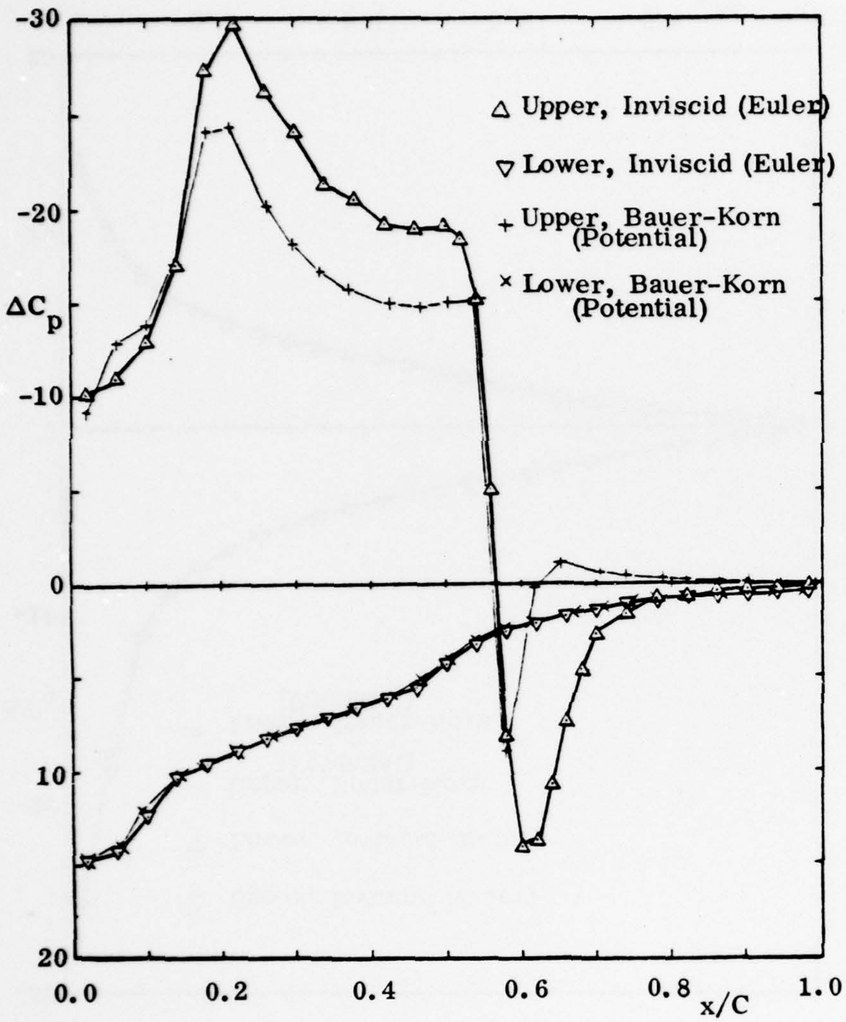


Figure 10. Pressure Excursions in Quasi-Steady Flow, Mach 0.721, $\alpha = 0.00°$, Unrestricted Stream

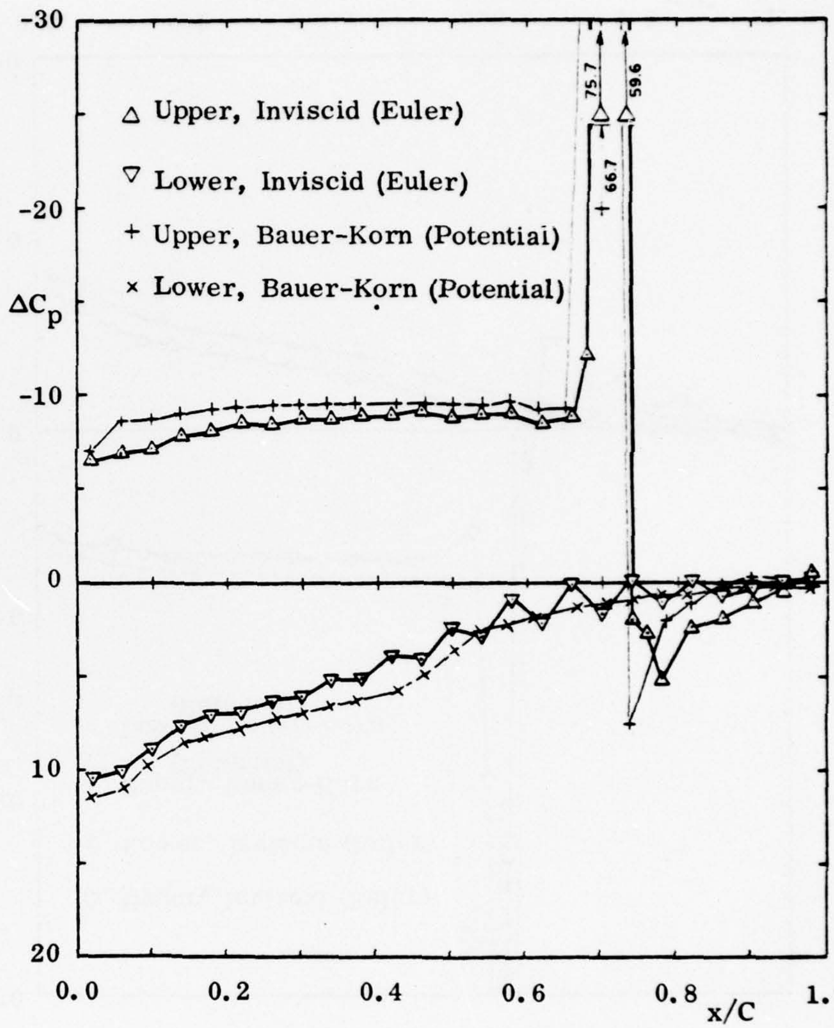


Figure 11. Pressure Excursions in Quasi-Steady Flow, Mach 0.744, $\alpha = 0.85°$, Unrestricted Stream

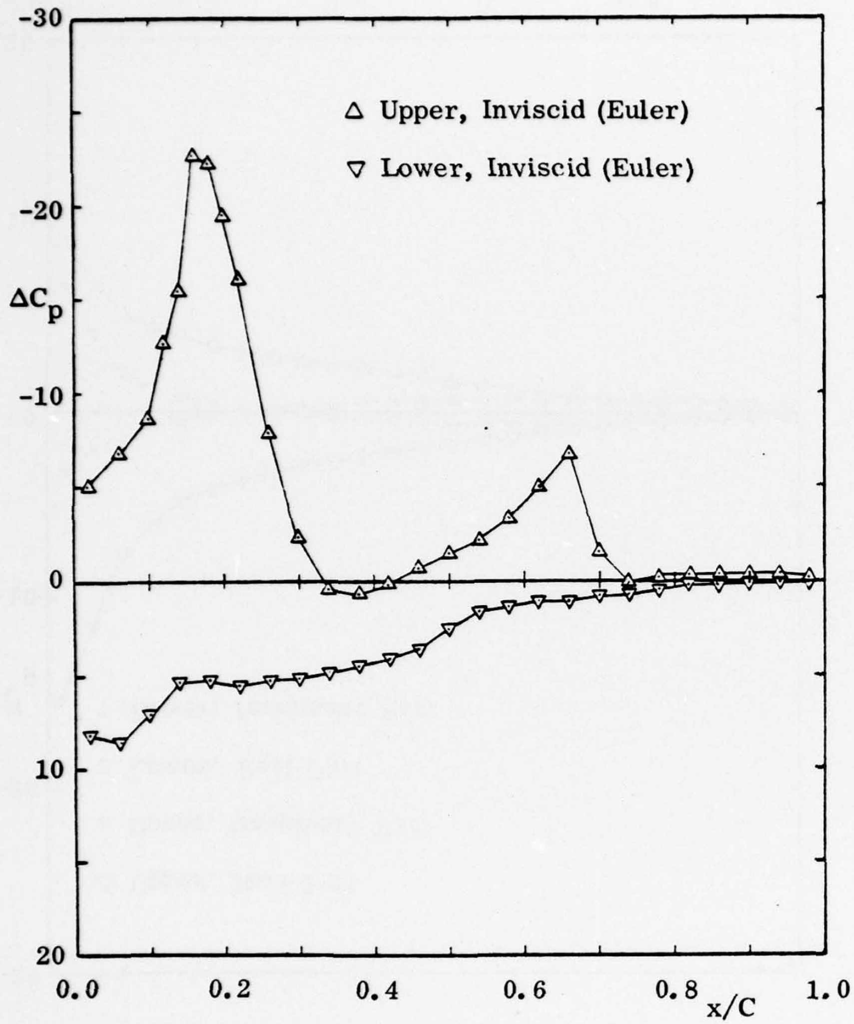


Figure 12. Pressure Excursions in Quasi-Steady Flow, Mach 0.744, $\alpha = 0.85°$, Slotted Tunnel Walls at y = ±1.53 Chords

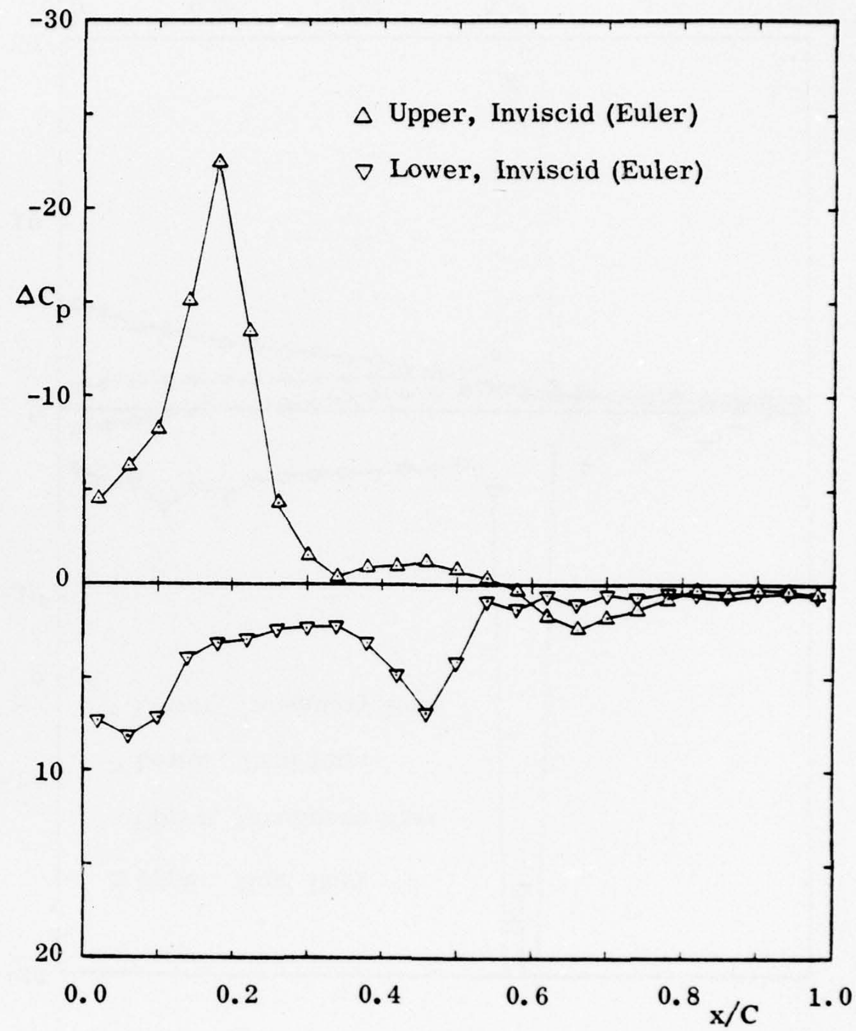


Figure 13. Pressure Excursions in Quasi-Steady Flow, Mach 0.744, $\alpha = 0.85°$, Free Jet Surfaces at y = ±1.53 Chords

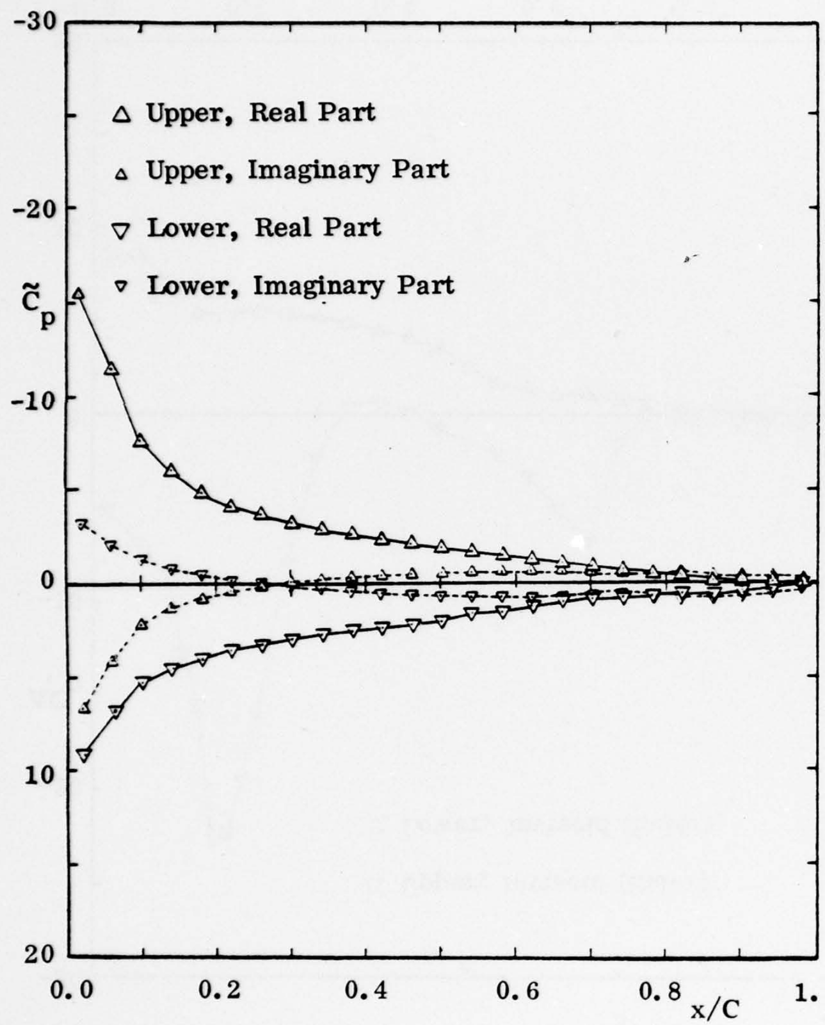


Figure 14. First Harmonics of Unsteady Pressure Excursions, Mach 0.500, $\bar{\alpha} = 0.85°$, k = 0.263, Unrestricted Stream

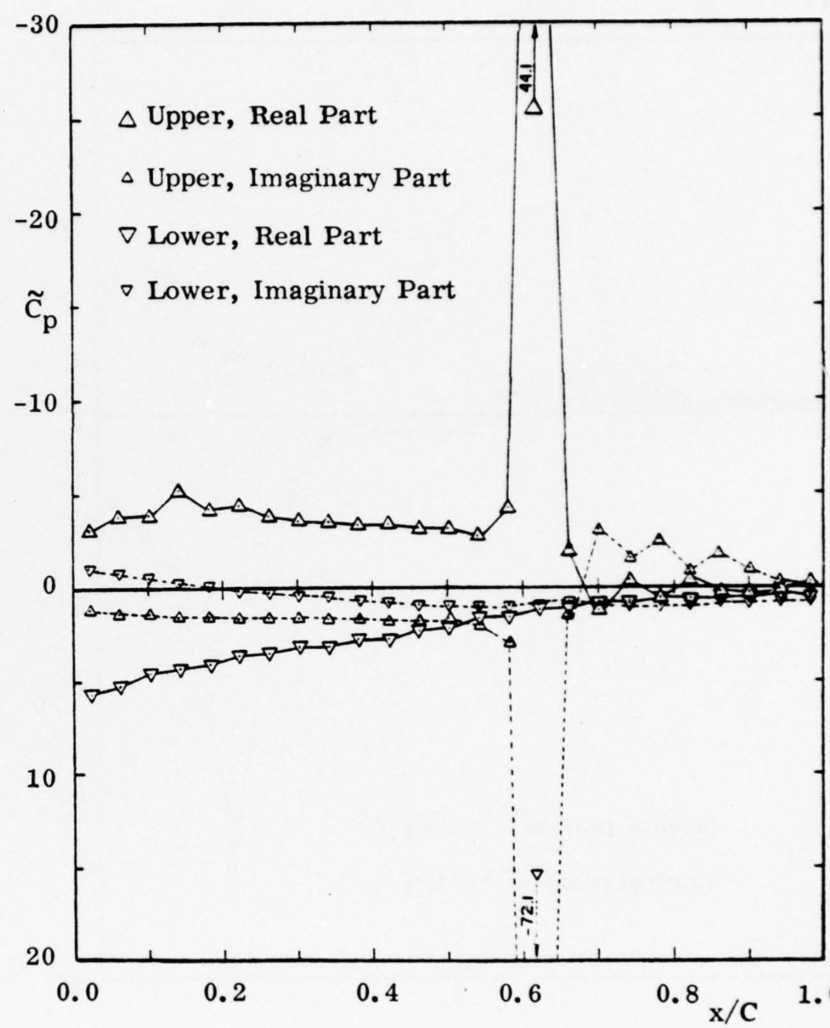


Figure 15. First Harmonics of Unsteady Pressure Excursions, Mach 0.700, $\bar{\alpha} = 3.00°$, k = 0.192, Unrestricted Stream

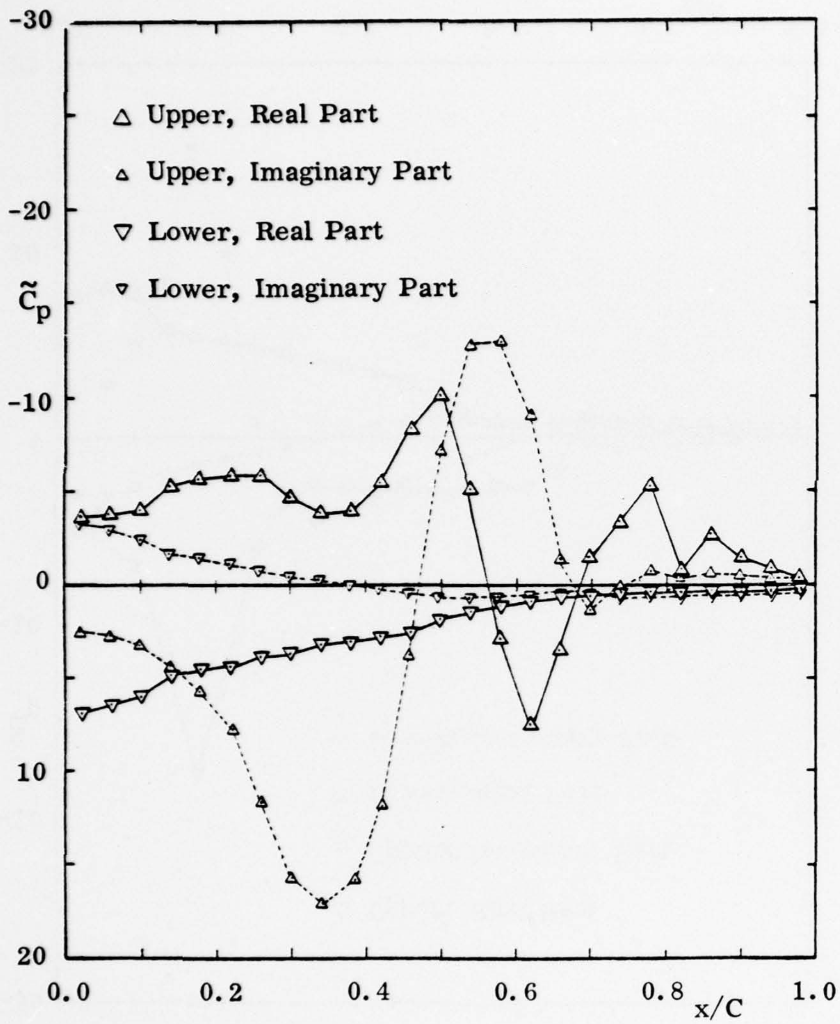


Figure 16. First Harmonics of Unsteady Pressure Excursions, Mach 0.721, $\bar{\alpha} = 0.00°$, k = 0.189, Unrestricted Stream

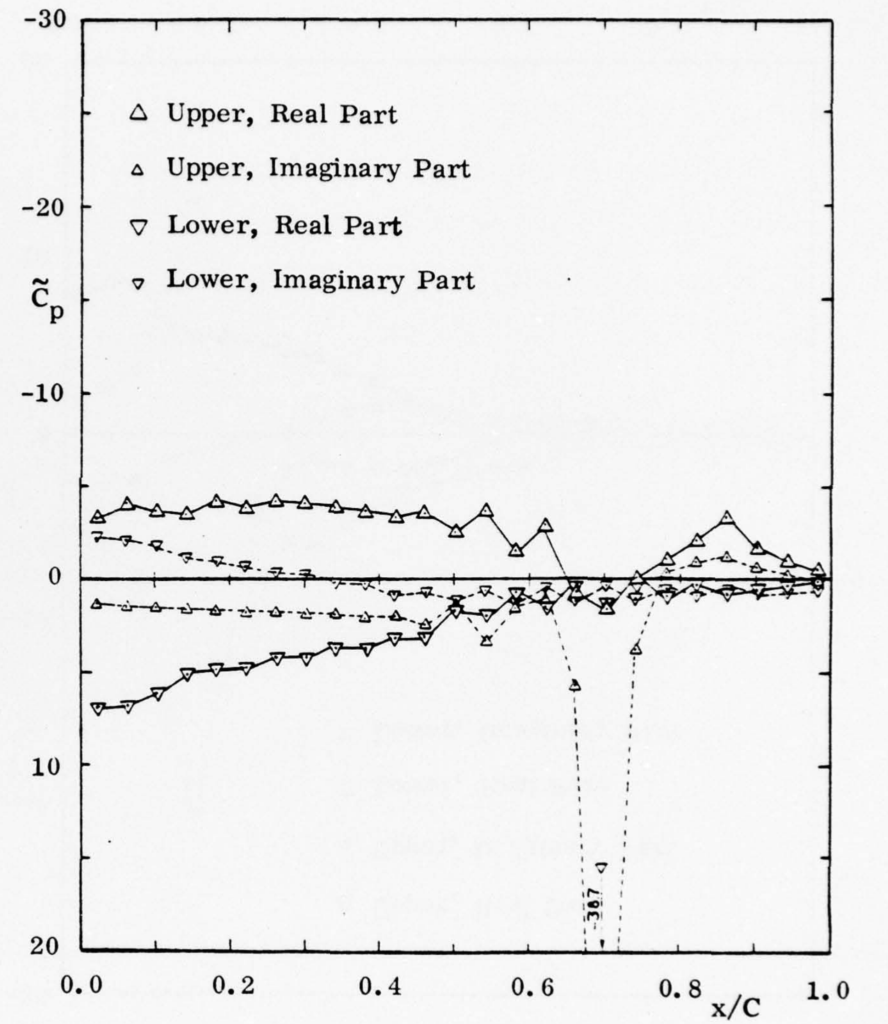


Figure 17. First Harmonics of Unsteady Pressure Excursions, Mach 0.744, $\bar{\alpha} = 0.85°$, k = 0.181, Unrestricted Stream

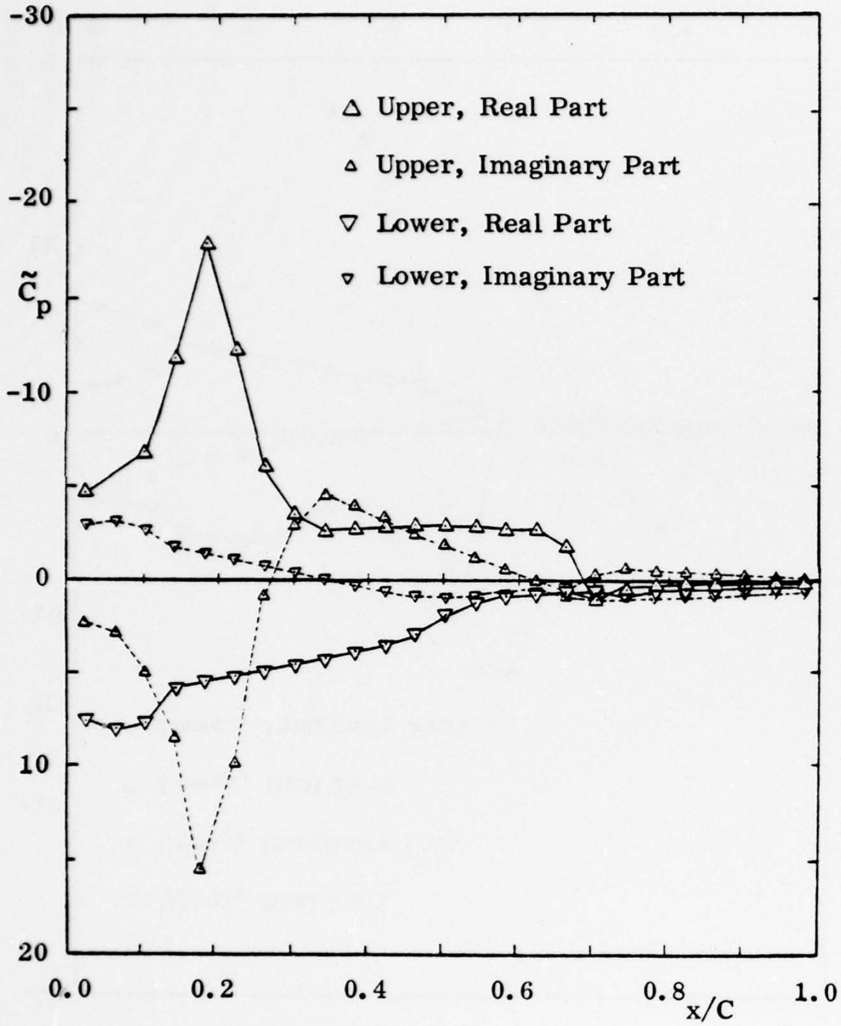


Figure 18. First Harmonics of Unsteady Pressure Excursions, Mach 0.744, $\bar{\alpha} = 0.85°$, k = 0.181, Slotted Tunnel Walls at y = ±1.53 Chords

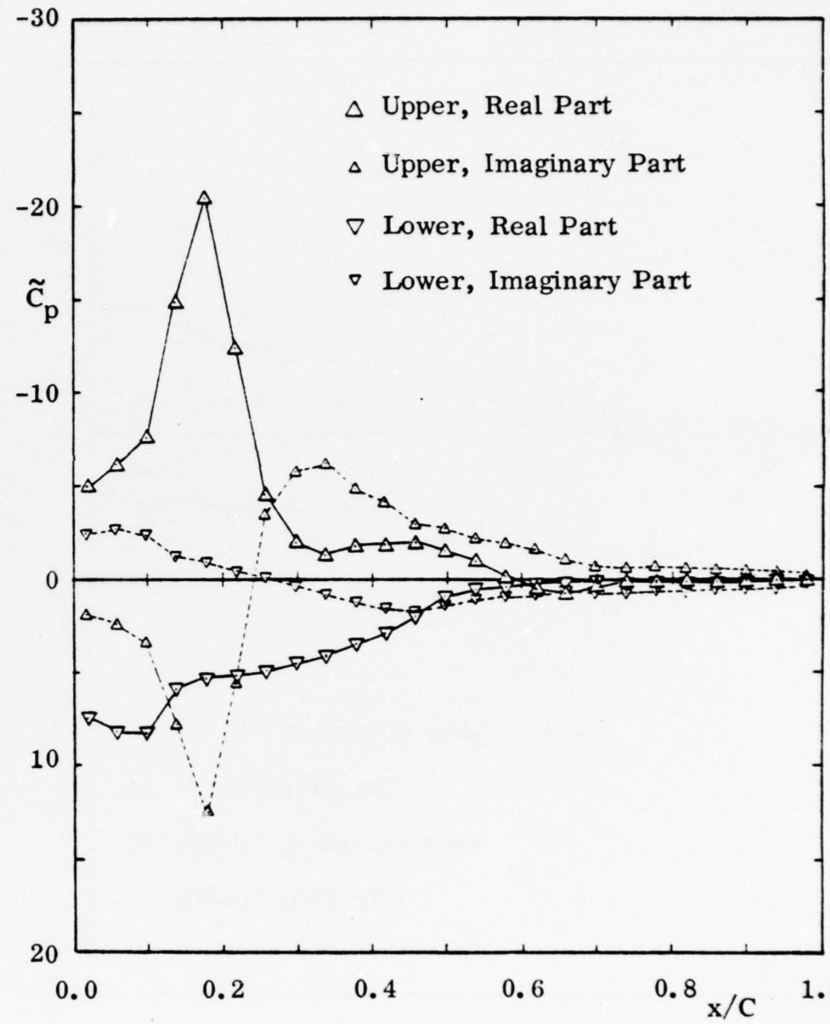


Figure 19. First Harmonics of Unsteady Pressure Excursions, Mach 0.744, $\bar{\alpha} = 0.85°$, k = 0.181, Free Jet Surfaces at y = ±1.53 Chords

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Inviscid transonic flows over the NLR 7301 airfoil were calculated using a program based on the unsteady Euler equations. The six cases treated include a subsonic flow, a near-shockless supercritical flow and some supercritical flows with shocks. In each case, steady flows and unsteady flows for pitching oscillations with 0.5 degree amplitude and reduced frequency ($k \equiv \omega C/2U_{\infty}$) of approximately 0.2 were calculated.

omega sub infinity

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